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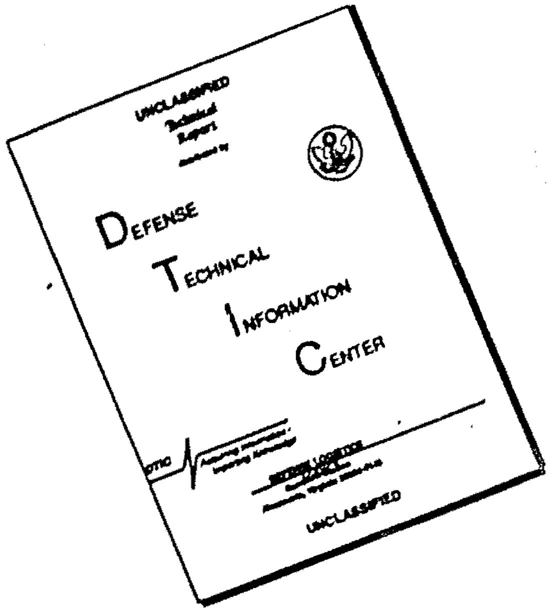


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DEPARTMENT OF AERONAUTICAL ENGINEERING

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DEPARTMENT OF THE NAVY
BUREAU OF AERONAUTICS

Contract NOas 53-817-c

FC

# COMBUSTION INSTABILITY

# IN

# LIQUID PROPELLANT ROCKET MOTORS

Twenty-Sixth Quarterly Progress Report

For the Period 1 August 1958 to 31 October 1958

Aeronautical Engineering Report No. 216-z

Prepared by David T. Harrje
D. T. Harrje, Project Engineer

Approved by [signature]
L. Crocco, Professor-in-Charge

23 February 1959

PRINCETON UNIVERSITY

Department of Aeronautical Engineering

CONTENTS

| | | Page |
|---|---|---|
| | TITLE PAGE | 1 |
| | CONTENTS | 2 |
| I. | SUMMARY | 3 |
| II. | INTRODUCTION | 4 |
| | A. Object | 4 |
| | B. History | 4 |
| III. | APPARATUS | 15 |
| | A. Test Stand No. 1 | 15 |
| | B. Test Stand No. 2 | 16 |
| | C. Test Stand No. 3 | 18 |
| | TABLE I | 20 |
| | FIGURES | |

## I. SUMMARY

Window motor runs on the standard injector (Test Stand No. 1) were completed and provided qualitative information as to the velocity distribution and combustion distribution in the chamber. Evidence was found of recirculation patterns arising from fuel droplet momentum transfer at low mixture ratios. The "standard" like-on-unlike injector was shown to have a varying [illegible] angle with mixture ratio which was partially responsible for the velocity patterns within the chamber.

The showerhead injector was investigated on the stability limits test stand (No. 2). The lower length limit of the fundamental mode instability region was found to occur at a considerably greater cylindrical chamber length than that found for the like-on-unlike type injectors previously tested. Preliminary values of the time lag parameters, $\tau$ and $n$, were determined although no check could be provided using the long length limits. These values indicated an increase in the sensitive time lag values due to the effect of slower droplet breakup.

Stability limits tests were also conducted using the like-on-like injector. With this injector the first appearance of combustion instability occurred at even greater lengths. Because of the nature of the injector the assumptions of the present calculation technique derived from the combustion instability theory do not apply and thus the time lag parameters could not be determined at this time.

The 100 lb. thrust level tests on the transverse mode test stand have indicated the existence of several types of transverse instability on two diameter chambers. Mixture ratios were varied between 1.0 and 2.2 for the majority of tests and pressures were maintained at 150, 500 and [illegible]00 psi. In all cases the orientation of the spray fans produced by the like-on-unlike injection was in the radial direction.

## II. INTRODUCTION

### A. Object

BuAer Contract NOas 52-713-c was undertaken as a part of the Jet Propulsion Research Program of the Department of Aeronautical Engineering at Princeton to "conduct an investigation of the general problem of combustion instability in liquid propellant rocket engines. This program shall consist of theoretical analyses and experimental verification of theory. The ultimate objective shall be the collection of sufficient data that shall permit the rocket engine designer to produce power plants which are relatively free of the phenomena of instability. Interest shall center in that form of unstable operation which is characterized by high frequency vibrations and is commonly known as 'screaming'."

### B. History

Interest at Princeton in the problem of combustion instability in liquid propellant rocket motors was given impetus by a Bureau of Aeronautics Symposium held at the Naval Research Laboratory on the 7th and 8th of December, 1950. This interest resulted in theoretical analyses by Professors M. Summerfield and L. Crocco of this center.

Professor Summerfield's work, "Theory of Unstable Combustion in Liquid Propellant Rocket Systems" (JARS, September 1951), considered the effects of both inertia in the liquid propellant feedlines and combustion chamber capacitance with a constant combustion time lag, and applied to the case of low frequency oscillations (up to about 200 cycles per second) sometimes called "chugging."

Professor Crocco advanced the concept of the pressure dependence of time lag in mid-1951: his paper "Aspects of Combustion Stability in Liquid Propellant Rocket Motors:" (JARS, November 1951 and Jan.-Feb. 1952), presents

the fundamentals resulting from this concept, and analyzes the cases of low frequency instability with bipropellants and high frequency instability with combustion concentrated at the end of the combustion chamber.

Desiring to submit the concept of a pressure dependent time lag to experimental tests, a preliminary proposal was made by the University to the Bureau of Aeronautics in the summer of 1951, and following a formal request, a revised proposal was submitted which resulted in Contract NOas 52-713-c.

Analytical studies with concentrated and distributed combustion had been carried on in the meantime under Professor Crocco's direction and within the sponsorship of the Guggenheim Jet Propulsion Center by S. I. Cheng and were issued as his Ph.D. thesis, "Intrinsic High Frequency Combustion Instability in a Liquid Propellant Rocket Motor," dated April, 1952.

Time was devoted in anticipation of the contract, during the first third of 1952, to constructing facilities, securing personnel, and planning the experimental approach.

During the first three month period of the contract year, personnel and facilities at the new James Forrestal Research Center were assigned, and the initial phases of the experimental program were planned in some detail.

A constant rate monopropellant feed system was designed and preliminary designs of the ethylene oxide rocket motor and the instrumentation systems were worked out. Special features of the projected systems included a flow modulating unit to cause oscillations in propellant flow rate, a water-cooled strain-gage pressure pickup designed for flush mounting in the rocket chamber, and several possible methods for dynamic measurement of an oscillating propellant flow rate.

Searches were made of the literature for sources of information on combustion instability and ethylene oxide, and visits to a number of activities

working on liquid propellant rocket combustion instability problems were made for purposes of familiarization with equipment and results.

The basic precepts of Crocco's theory for combustion instability were reviewed, and detailed analyses made for specific patterns of combustion distribution. Operational tests and calibration of the Li-Liu pressure pick-up proved the value of the design, although failure of the pickup under "screaming" rocket conditions showed the necessity for modification of the cooling system.

Construction of the monopropellant test stand and rocket motor were completed. Modifications were made to the Li-Liu pressure pickup to provide for higher permissible heat-transfer rates in order that it be satisfactory for use under "screaming" conditions in a bipropellant rocket motor. Construction and preliminary testing of the hot-wire flow phasemeter and its associated equipment were completed.

A new contract, NOas 53-817-c, dated 1 March 1953, was granted by the Bureau of Aeronautics to continue the program originally started under NOas 52-713-c. Monopropellant rocket tests were started under this new contract, and shakedown operations were completed. It was found that decomposition of ethylene oxide could not be attained with the original motor design despite many configuration changes, and it was decided to avoid a long and costly development program by operating the "monopropellant" motor with small amounts of gaseous oxygen. The required limits of oxygen flow rate were determined at several chamber pressures and it was indicated that the oxygen would probably have a negligible effect on performance when compared to the effect of ethylene oxide flow rate modulation. Preliminary tests with flow rate up to 100 cps were performed for the purposes of system checkout, using interim AC amplifiers in lieu of the necessary precision instruments.

The time constant of the hot-wire liquid flow phasemeter was found to be 0.15 milliseconds and preparations for instantaneous flow calibration were made. A test rig was constructed for this purpose.

A bipropellant rocket system using liquid oxygen and 100% ethyl alcohol was designed on the basis of monopropellant operational experience, incorporating an adjustable-phase flow modulating unit in both propellant flow lines. Injector design was based on a configuration used extensively by Reaction Motors, Inc.

Operation of the monopropellant system was performed with flow modulation at frequencies up to 120 cps, using a composite instrumentation system to measure mean values, amplitudes of oscillation, and phase difference between injector and chamber pressures. Analysis of the results of this program demonstrated approximate adherence to the pressure-time lag relationship used in Crocco's original theoretical treatment. Accuracy of the measurements was not adequate to provide a detailed check of the theory, however, so an instrument refinement and development program was initiated.

The Mittelmann electromagnetic flowmeter proved unsatisfactory due to equipment malfunctions, and had to be abandoned as a possible means of measuring instantaneous flow rates. The Li dynamic flowmeter also experienced a number of mechanical failures, and it was decided to concentrate all flow-phase measurement effort on the hot-wire, which, of all flowmeters tested, appeared to show the most reliable results.

Shakedown operation of the bipropellant rocket chamber at 300 and 600 spi was completed, and steady state mixture-ratio tests were run satisfactorily. All components of the bipropellant test stand were checked out including the flow modulating unit, which was operated at speeds up to 12,000 rpm. The only difficulty encountered was minor cracking of the first

two injectors due to stress concentrations at a sharp-cornered groove. No further trouble ensued after correcting this condition.

Dynamic water calibrations were run to determine operating limits of each bipropellant injector, each set of cavitating venturis, and each set of flow modulating pistons, at frequencies of 50 to 200 cycles per second.

The instrumentation refinement program was completed with the exception of the phase-measuring procedure and a few other minor features. The modified data recording systems (both transient and steady-state) were used successfully on the dynamic flow calibrations and on several rocket tests.

Bipropellant tests with flow modulation were run at chamber pressures of 300 and 600 psi and flow modulating frequencies of 50 to 200 cycles per second. Further heat-transfer failures of the Li-Liu pressure pickups at the higher chamber pressure levels necessitated further investigation and improvement of the cooling system of the pickups, but their performance continued to be generally satisfactory.

The liquid hot-wire flow phasemeter was checked for thermal response and found to be satisfactory within the accuracy of the test data. Preliminary liquid phase-lag calculations for the bipropellant injector demonstrated the need for a detailed calculation and experimental check of this quantity.

A new sectioned bipropellant chamber was designed for maximum flexibility in establishing the experimental limits of stability criteria, and also for determination of chamber parameter effects on instability.

The first detailed series of bipropellant flow-modulated runs was made at 300 psi, but the results were found to contain excessive scatter. Part of this was traced to zero drift of the pressure pickups, but even when gage photographs were used for the steady-state component, it was found that

liquid oxygen compressibility was excessive. Sensitivity checks of the pressure pickups demonstrated satisfactory performance.

Additional pickup cooling capacity was sufficient to provide satisfactory operation under screaming conditions at 600 psi chamber pressure. The use of a cooled adapter around the pickup mounting threads had no apparent effect on either heat transfer or zero drift. The phase measuring system to be used on the rocket data was designed, and a prototype circuit constructed.

The hot-wire was found to be a satisfactory instrument for the measurement of transient flow rates in liquid lines, but results of detailed water calibrations showed it to be very sensitive to cavitation, bubbles, etc.

Design and construction of the sectioned motor designated for limits of stability testing were completed, and installation on the old monopropellant test stand (henceforth called Bipropellant Stand No. 2) was made. Construction of instrumentation required for axial pressure measurements and run history records on this chamber was completed. A method for measuring heat transfer from the chamber by means of the thermocouples was established.

The 300 psi series of flow-modulated bipropellant runs was repeated using a full flow precool period to eliminate oxygen vapor. Measurements of the chamber transfer function were suitable for determination of the time lag and associated parameters, although the existence of mixture ratio variation introduced some degree of error into the computed results. Runs at 450 psi chamber pressure were also made.

A new method of obtaining transient flow rates by measuring the instantaneous momentum efflux from the bipropellant injector was devised, and construction of experimental equipment completed.

The limits of stability test motor was operated successfully on Bipropellant Stand No. 2, proving the suitability of its design. Several

runs were also made with high-sensitivity, steady-state pressure pickups to determine the axial combustion distribution in the chamber, but these first results were not successful.

The method for measuring heat transfer with wall thermocouples was calibrated, but only very small signals were obtained. It was found that the method of thermocouple seating used was not satisfactory.

The Li-Liu pressure pickup was modified according to Princeton's suggestions, and initial tests of the modified version indicated no thermal drift under screaming conditions at 600 psi.

Apparatus for measurement of exhaust nozzle impedance to high frequency oscillations was designed, and construction was begun. Numerical calculations and initial experimental studies to check out the facility were completed. It was found necessary to increase motor power and revise the hot-wire anemometer system.

A refined series of flow-modulated test data was run at 300 psi with modulating frequencies from 50 to 210 cycles per second. It was found that the phase and amplitude of mixture ratio oscillations were interdependent, but the variation in mixture ratio was vastly improved over earlier tests. It was believed that a second-order correction would be required to achieve constant mixture ratio. It was also found that an extension of range of the flow modulating unit to 250 cps was required to obtain the necessary asymptotes, and this modification was made.

Strain beam equipment for measuring instantaneous flow rate by the momentum method was installed on Rocket Test Stand No. 1, and calibrations showed it to be adequate for the required measurements.

The Third Conference on Rocket Combustion Instability took place at Princeton on October 18 and 19, 1955. The Princeton group had papers

presented by Crocco, Grey, Matthews and Scala.

The refined low-frequency chamber analysis, which eliminates a number of the limiting assumptions present in the original analysis used on all experimental data to date, was completed.

Longitudinal stability limits were measured and checked on Bipropellant Test Stand No. 2 using four exhaust nozzles: a short linear nozzle, a short conical nozzle, a long linear nozzle, and a long conical nozzle. Theoretical limits of stability were computed for the two linear nozzles and found to compare very favorable with the experimentally-determined values, thus substantiating the major portions of the time-lag theories in at least a preliminary way. It was also found that the long nozzles both depressed the screaming frequency and markedly decreased the tendency to scream, in accordance with theoretical predictions.

Theoretical analysis of the combustion chamber in which mixture ratio changes are permitted to occur was carried out and the quantitative criteria for intermediate-frequency instability established for one motor configuration.

Flow-modulated test data at chamber pressures of 300 psi, 450 psi, and 600 psi were completed, with the smallest possible mixture ratio variation obtainable using existing equipment. An average of 5 to 10 runs were required for each data point of reasonably satisfactory mixture-ratio variation.

Stability limit tests were run at the three chamber pressures and the results compared quite favorably with theory. This is considered to be a conclusive check of the theory for at least these test conditions.

Studies of the method of measuring heat transfer using thermocouples were completed. It was concluded that the calibration apparatus required both cooling and a high-energy heat source, but application of alternate methods

of analysis to the experimental data proved quite satisfactory.

The Statham pressure pickups for use in measuring combustion distributions on Bipropellant Test Stand No. 2 were installed with a system of solenoid valves to completely isolate the Statham pressure pickups from the chamber during severe shutdown transients. This new system for measuring combustion distribution was operated with some success, but failure of all pickups occurred during an AC power failure to the test stand. An interim system using high-pressure manometers was constructed as a replacement. Measurements of velocity distribution were made with the manometers at the three chamber pressure levels. Manometer operation was satisfactory, but it was found that the chamber flow was not one-dimensional and thus the axial pressure distribution method was not applicable. A window motor was designed and constructed for direct measurement of the velocity distribution.

The Ampex tape recorder was converted to seven channels, completing the entire instrumentation system for time-lag measurements.

Design, construction, and shakedown of a closed-loop, servo-controlled flow modulating unit to provide automatic mixture ratio regulation were completed. Use of this unit on rocket tests was found to be completely satisfactory.

Stability limit tests were made to determine the effect of mean mixture ratio on the longitudinal mode. A device to introduce high-pressure longitudinal shocks into the chamber was designed to determine limits of nonlinear stability. A number of injectors for use with different propellant combinations were also designed.

Theoretical analysis of the transverse mode of high-frequency instability was completed and a composite chamber for transverse mode tests was designed.

A technical report covering the results of time lag measurement studies was published, highlighting further improvements which could be made for the purpose of accuracy refinement.

Tests to determine longitudinal stability limits for sudden, finite-amplitude pressure changes were run at 300 psi, and indicated only small departure from the spontaneously-initiated stability limits determined previously. The use of chamber-pressure and mixture-ratio control on these tests was considered advisable due to some inconsistencies attributable to uncontrolled variations in these characteristics.

Errors contributing to inaccuracies in the time lag measurement were found and eliminated, including poor AC calibration standards and poor oxygen density control. Investigation of resonant oscillations in the feed-line was completed with the determination that the bubble in the cavitating venturi had been causing the difficulty and could be eliminated by optimum operation of the cavitating venturi.

An automatic chamber pressure and mixture ratio control system was designed, installed, and tested. It has been made available for all stability limit test data in order to quarantee reproducibility of run information.

A technical report covering the theory of the intermediate frequency and transverse-mode instability was published. Additional computations to further catalog the transverse-mode behavior from a theoretical point of view were initiated.

Complete verification of the nozzle admittance theory used in all the theoretical-experimental comparisons was established by a series of indirect tests in which the dynamic behavior of a simulated rocket chamber was experimentally checked against theoretical predictions.

The NARTS large-scale supplementary test program to be conducted in conjunction with the transverse-mode instability studies has been

approved by the Bureau, and a preliminary schedule has been set up.

The relationship between the modulus and phase of the chamber transfer function was determined on the 6 inch rocket motor on Test Stand No. 1. The frequency of flow pulsations used ranged from 80 to 260 cps. Data scatter was held to a minimum through improved experimental techniques.

Exploratory testing on the transverse mode hardware has covered chamber diameters ranging from 3 to 9 inches with chamber pressures at the 150 and 500 psi levels. No transverse instability was measured under these conditions. This injector design was modified to the spud-type which allows reorientation of the impingement fan. With the injector spray fan oriented in the radial direction definite regions of transverse instability have been recorded.

Using the standard injector, the extreme length stability tests were completed on Test Stand No. 2 for mixture ratios ranging from .8 to 3.5 using alcohol and liquid oxygen. Using the Crocco Instability Theory the long length limits of the fundamental mode instability contour have been accurately predicted and $\tau$ and $n$ have been determined.

Using the 2.0 mixture ratio injector, the accuracy of Crocco Instability Theory has been further verified through the prediction of the long length stability limits of the fundamental mode.

## III. APPARATUS

### A. Test Stand No. 1

The window motor runs to determine the combustion distribution from the gas velocity in a liquid propellant rocket motor have been completed and are presented here for the "standard" injector using liquid oxygen and ethyl alcohol.

These data were obtained from analysis of film loop records using the General Radio Camera to study the combustion in a six inch cylindrical length rocket motor incorporating a lucite window along the axis. The streak photographs of the combustion at 1" and 3" distances from the injector face with varying mixture ratio provided the data shown in Figure 1. The theoretical velocity, included for comparison, was calculated assuming completed combustion and a uniform velocity profile, a condition approached with high Reynold's Number flow.

From the velocity variation with mixture ratio at the 1" station, it is seen from a comparison with the theoretical curve that combustion has been completed before reaching that point. The experimental velocities are higher than theoretical values because the velocity measurements were made along the rocket chamber axis and a near uniform velocity profile was not attained. These velocity profiles are shown in Figure 2, where for the 2.0 mixture ratio case velocities were determined at the 1/2" and 1" distances from the axis. The profile was completed by determining the velocities required to provide for the total flow present with complete combustion.

The velocity variation with mixture ratio and radius from the rocket axis for the low mixture ratio tests as observed at the 3" station are of interest. Although at high mixture ratios the velocities measured were close to the velocity values found at the 1" station, at the low mixture ratios velocities along the rocket axis were found to be considerably

higher. Using a velocity profile similar to that determined experimentally it is seen in Figure 2 that a region of low velocity would be required near the rocket chamber wall in order to attain the correct net flow. This "recirculation region" is due to several causes. The first, as illustrated in Figure 3, is that the angle of the resultant liquid stream changes with mixture ratio. During low mixture ratio operation the resultant streams converge at the center of the rocket motor (shown in Figure 4). Since the stoichiometric mixture ratio is 2.05 for this alcohol-oxygen system a considerable amount of fuel droplets remain after combustion has taken place and these droplets cause the velocity increase through momentum transfer. In the case of the high mixture ratio operation these droplets are no longer present and the direction of the injected liquid streams is toward the wall both of which tend to eliminate recirculation. Wall static pressure measurements have provided a confirmation to these results.

For the showerhead type injector shown in Figure 5, it was not possible to obtain useful data with the window motor technique. In order to obtain the necessary velocity distribution information with this rather slow burning injection system, the length of the stability limits rocket motor on Test Stand No. 2 was varied and the resultant characteristic velocity-length relationship determined. These results are presented with the Test Stand No. 2 data.

B. Test Stand No. 2

During this report period the type of injection used on the stability limits tests with ethyl alcohol and liquid oxygen was extended to include the showerhead and like-on-like injectors. Again, as in the case of the two variations of like-on-unlike injectors (1.4 and 2.0 design mixture ratio types), the mixture ratio was varied between approximately .5 and 4.0 for cylindrical

chamber lengths varying from 3 to 36 inches. The purpose for these tests was to determine to what extent the stability limits and time lag parameters were altered with changes in the method of injection using the same propellant combination. In the showerhead injector (Figure 5) droplet breakup was dependent only on the interaction of the liquid stream with the chamber combustion gases, while in the like-on-like pattern a spray of droplets was formed close to the injector face where the liquid streams impinge. Both types of rocket injection must depend on parameters such as recirculation velocities and droplet dispersal to bring the fuel and oxidizer droplets together. An interesting case of how the presence of combustion instability influences the droplet mixing and breakup is shown in the case of the showerhead injector in Figure 6. A sharp performance increase occurs with the presence of oscillating pressure and velocity in the combustion gases for the 1.5 and 1.2 mixture ratio cases. Without instability the 1.8 $\bar{r}$ curve illustrates the gradual performance increase expected. When steady conditions reoccur for the 1.5 $\bar{r}$ case at the 34 inch length little difference exists between the steady and nonsteady cases.

A portion of the fundamental mode instability contour for the showerhead injector is shown in Figure 7. The present hardware would not allow longer lengths to be run in this test series and therefore no check on the validity of the Crocco Theory using the long length instability limits could be made at this time. Compared with the like-on-unlike injector instability contours, the shorter length limits are seen to occur at much greater cylindrical lengths (22" versus 6"). Using the chamber Mach number ($\bar{u}$) as determined from characteristic velocity measurements at 4 inch chamber increments along the motor length as shown in Figure 9 it was possible to calculate a preliminary set of time lag parameter values. $\tau$ and $n$ are plotted in Figure 10. Comparison curves for the like-on-unlike injector

(2.0 mixture ratio) are shown in Figure 11. No large changes in either the shape or magnitude of the interaction index curve have occurred using the showerhead injector. However, the sensitive time lag has increased by a factor of approximately five. This change indicates the effect of the slower droplet breakup on the sensitive time lag.

In the case of the like-on-like injector (Figure 8) the first appearance of combustion instability occurs at a cylindrical length of 34 inches which is considerably greater than that found using the showerhead. Performance and hence chamber Mach number (Figure 9) are greater than the showerhead values at most chamber lengths due to earlier droplet breakup (except where instability was present in the showerhead case).

Because the combustion is distributed along the chamber length the assumption in the theory that energy is released in a concentrated region is no longer valid. Therefore, it is not possible to calculate the time lag parameters, $\bar{\tau}$ and $n$, at this time. Long length instability limits for this injector type would also be useful and will be determined. The effects of this type combustion distribution will be given further theoretical treatment.

C. Test Stand No. 3

Testing to date on the transverse mode rocket hardware has included 100 pound thrust level runs with two general motor configurations. One utilized the 6 inch diameter chamber, 5 inch injection diameter, 1.4 mixture ratio spuds with the fans oriented radially and using various nozzles to provide chamber pressures of 150, 500 and 900 psi. The other used a 9 inch chamber diameter, 8 inch injection diameter, 2.2 mixture ratio spuds with the fans oriented radially and nozzles to provide chamber pressures of 150 and 500 psi. A summary of the pressure oscillations produced in these tests is presented in Table I.

The wave shapes of the pressure oscillations produced were sinusoidal and of low amplitude in the majority of tests. The frequencies usually corresponded to a first tangential mode with a few instances of second tangential in the 9 inch chamber. The higher frequencies experienced with the 6 inch chamber, which occured at the higher mixture ratios and at the higher pressures, corresponded to the first radial mode.

It was also observed that at several of the low mixture ratio runs using the 6 inch chamber pressure oscillation amplitudes higher than the typical value of 10 psi rms or less were present. The wave shapes in these cases were also sinusoidal.

The next series of tests scheduled on the transverse mode test stand will provide thrusts at the 500 pound level. During such a test series it is planned to return to the "standard" fan orientation for the spuds (tangential fan direction) for stability comparison purposes.

TABLE I

Amplitudes of Transverse Oscillations at 100 lb. Thrust Level

| Test Conditions | | Oscillation Amplitudes (psi rms) | | |
|---|---|---|---|---|
| Chamber Pressure (psia) | Mixture Ratio (O/F) | First Tangential Mode | Second Tangential Mode | First Radial Mode |
| (a) Chamber diameter 9", Injection diameter 8", Design Mixture ratio 2.2 | | | | |
| 150 | 0.98 | 2 | - | - |
| " | 1.42 | 2 | - | - |
| " | 1.81 | - | - | - |
| " | 2.30 | - | - | - |
| | | | | |
| 500 | 0.95 | - | 4 | - |
| " | 1.34 | - | 12 | - |
| " | 1.65 | 8 | - | - |
| " | 2.37 | 12 | - | - |
| (b) Chamber diameter 6", Injection diameter 5", Design Mixture ratio 1.4 | | | | |
| 150 | 1.02 | 5 | - | - |
| " | 1.44 | 4 | - | - |
| " | 1.92 | 0 | - | - |
| " | 2.19 | 3 | - | - |
| " | 2.32 | 2.5 | - | - |
| | | | | |
| 500 | 1.04 | 50 | - | - |
| " | 1.22 | 25 | - | - |
| " | 1.40 | 0 | - | - |
| " | 1.42 | 3 | - | - |
| " | 1.45 | 7 | - | - |
| " | 1.57 | 0 | - | - |
| " | 1.74 | - | - | 5 |
| " | 1.88 | - | - | 3 |
| " | 1.92 | - | - | 3 |
| " | 2.12 | - | - | 1.5 |
| " | 2.20 | - | - | 2 |
| | | | | |
| 900 | 1.05 | 30 | - | - |
| " | 1.11 | 30 | - | - |
| " | 1.25 | - | - | 3 |
| " | 1.31 | - | - | 3 |
| " | 1.84 | - | - | 3 |
| " | 2.57 | - | - | 7 |

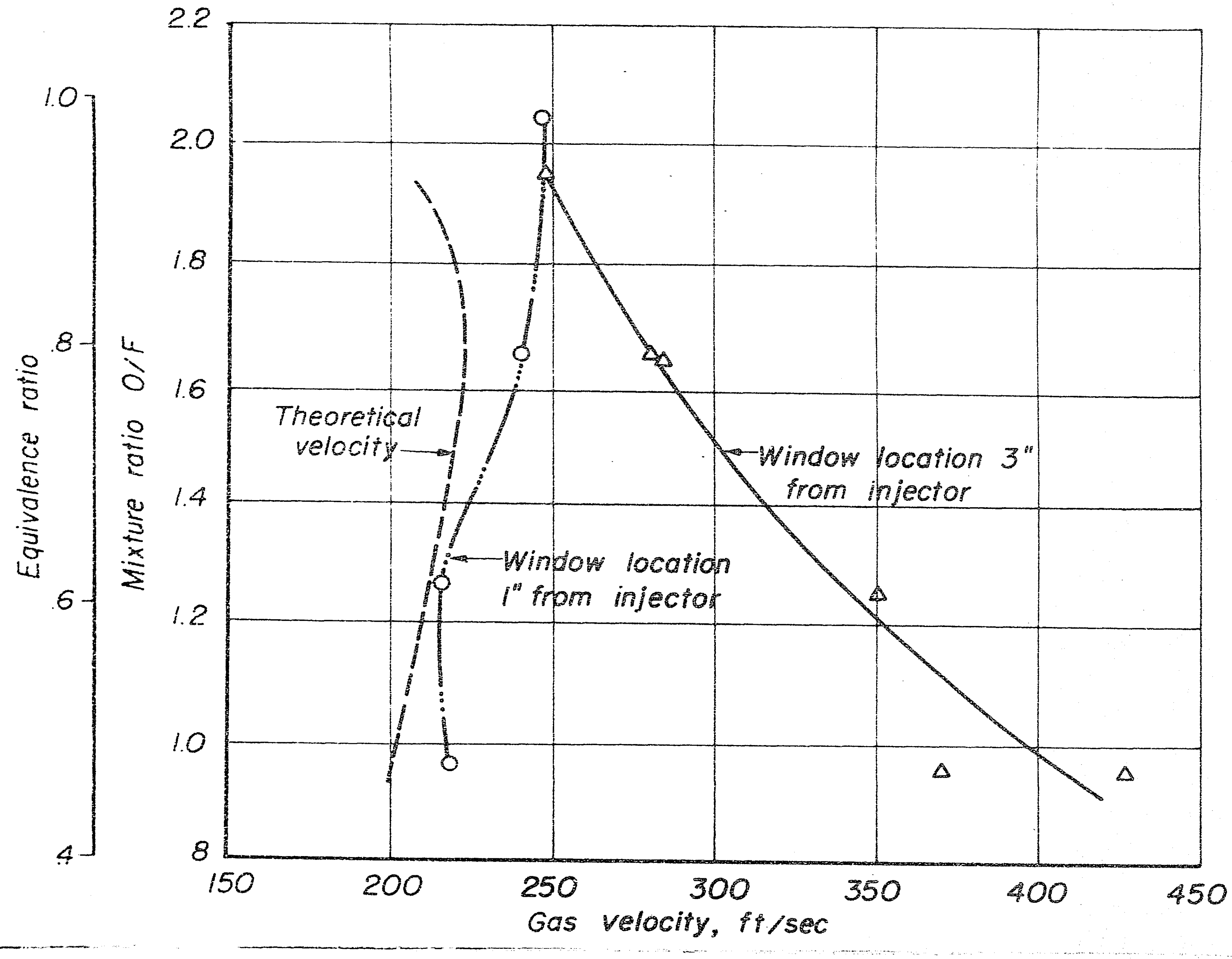
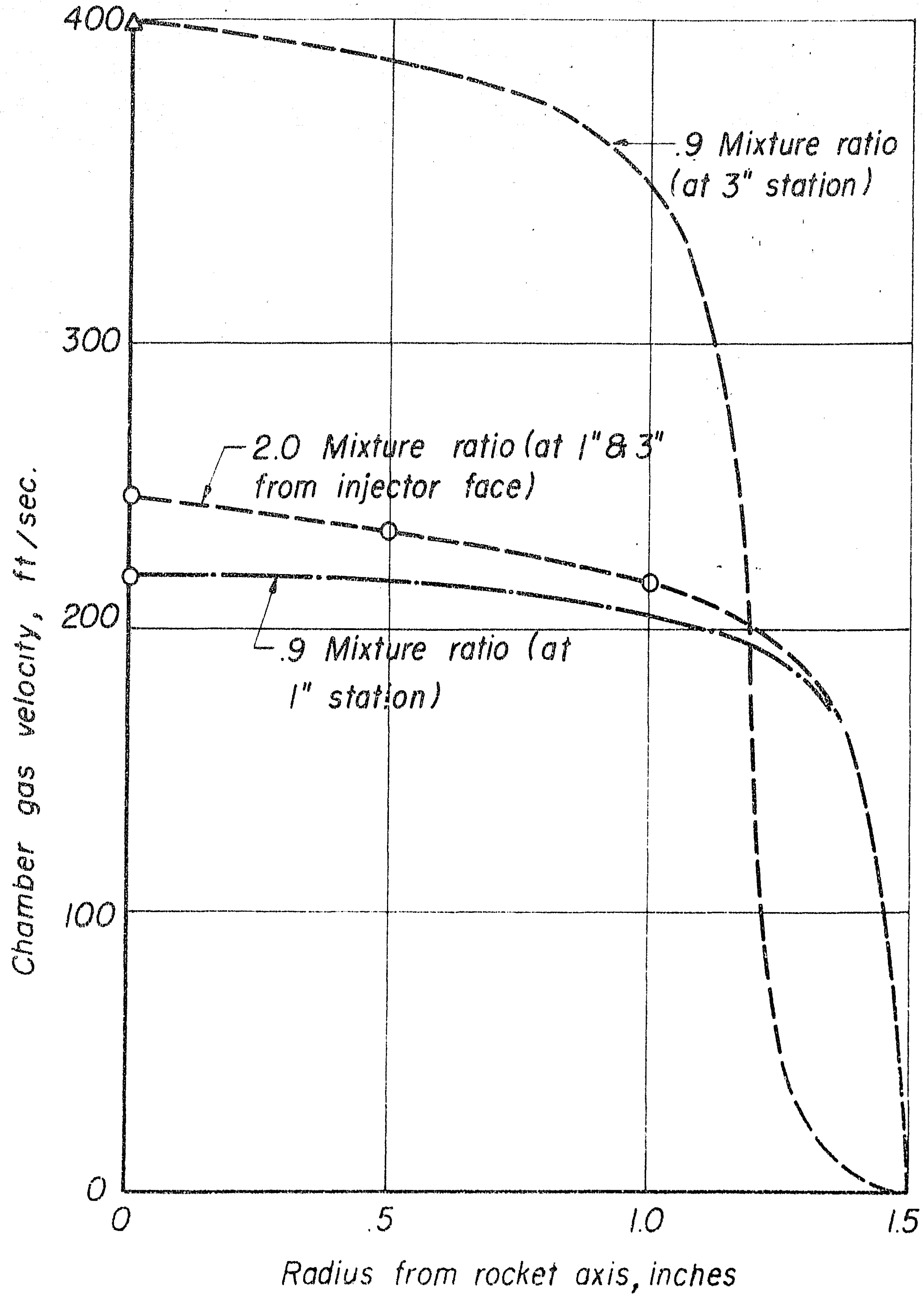

2.2
2.0
1.8
1.6
1.4
1.2
1.0
8
1.0
.8
.6
.4
Equivalence ratio
Mixture ratio O/F
150
200
250
300
350
400
450
Gas velocity, ft/sec
Theoretical velocity
Window location 3" from injector
Window location l" from injector

Figure I

Figure 2



400
300
200
100
0
.9 Mixture ratio
(at 3" station)
2.0 Mixture ratio (at 1" & 3"
from injector face)
.9 Mixture ratio (at
1" station)
Chamber gas velocity, ft/sec.
0
.5
1.0
1.5
Radius from rocket axis, inches

Figure 3

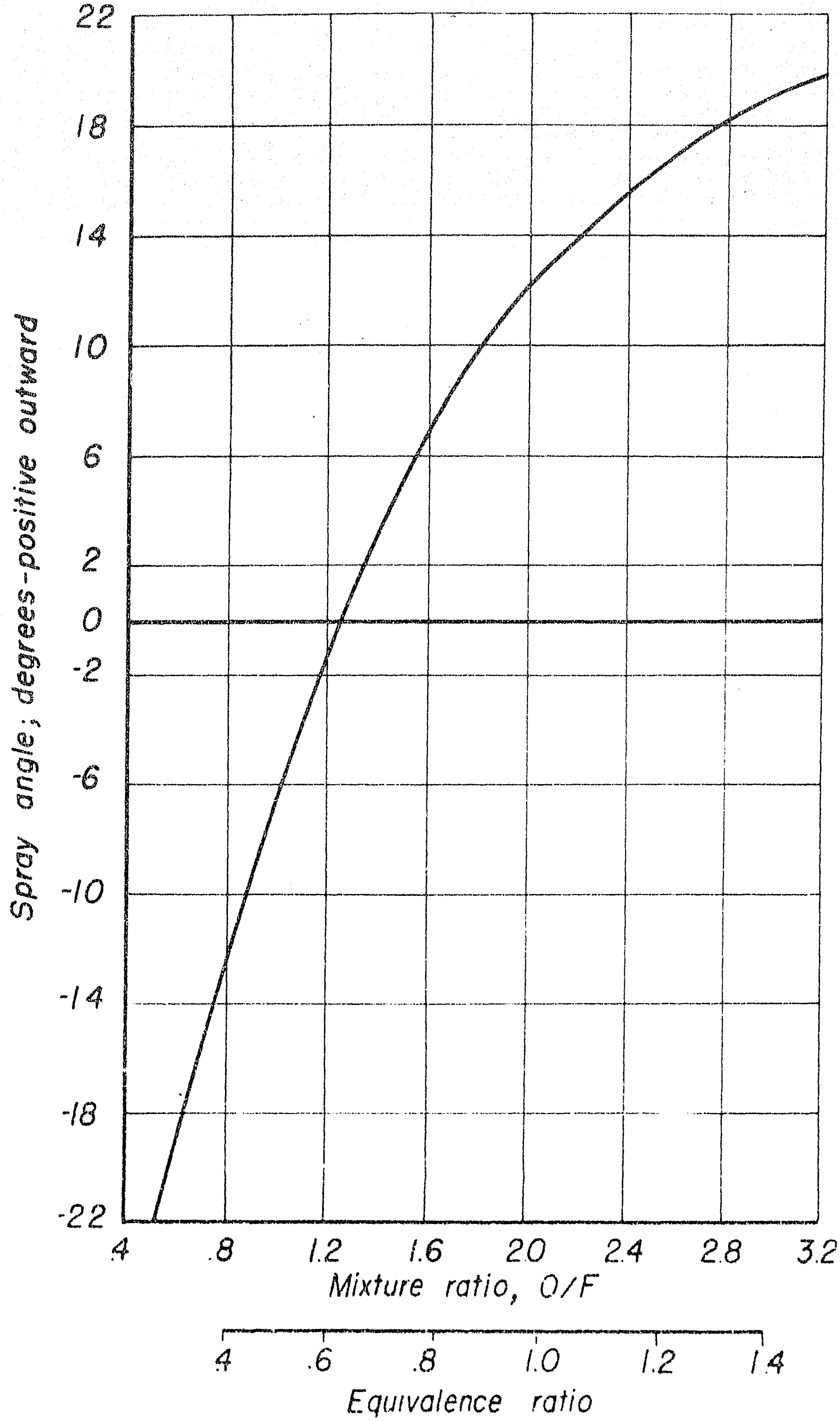



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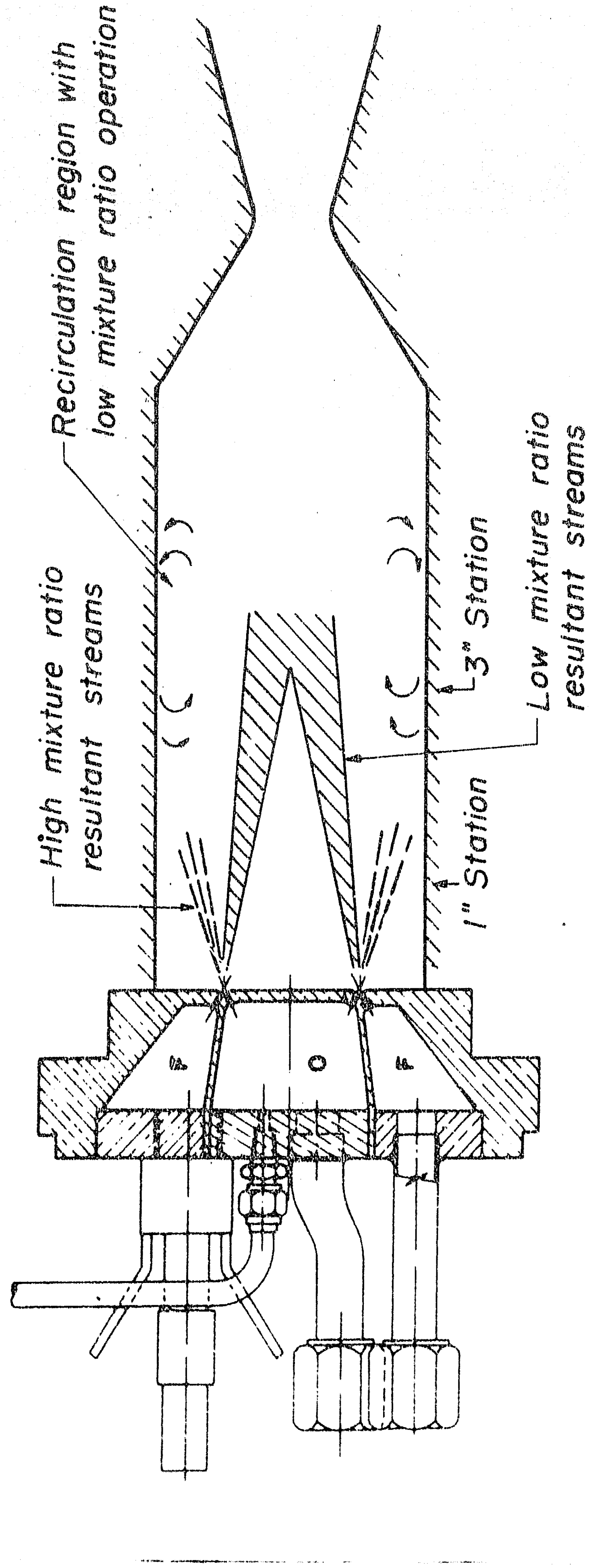
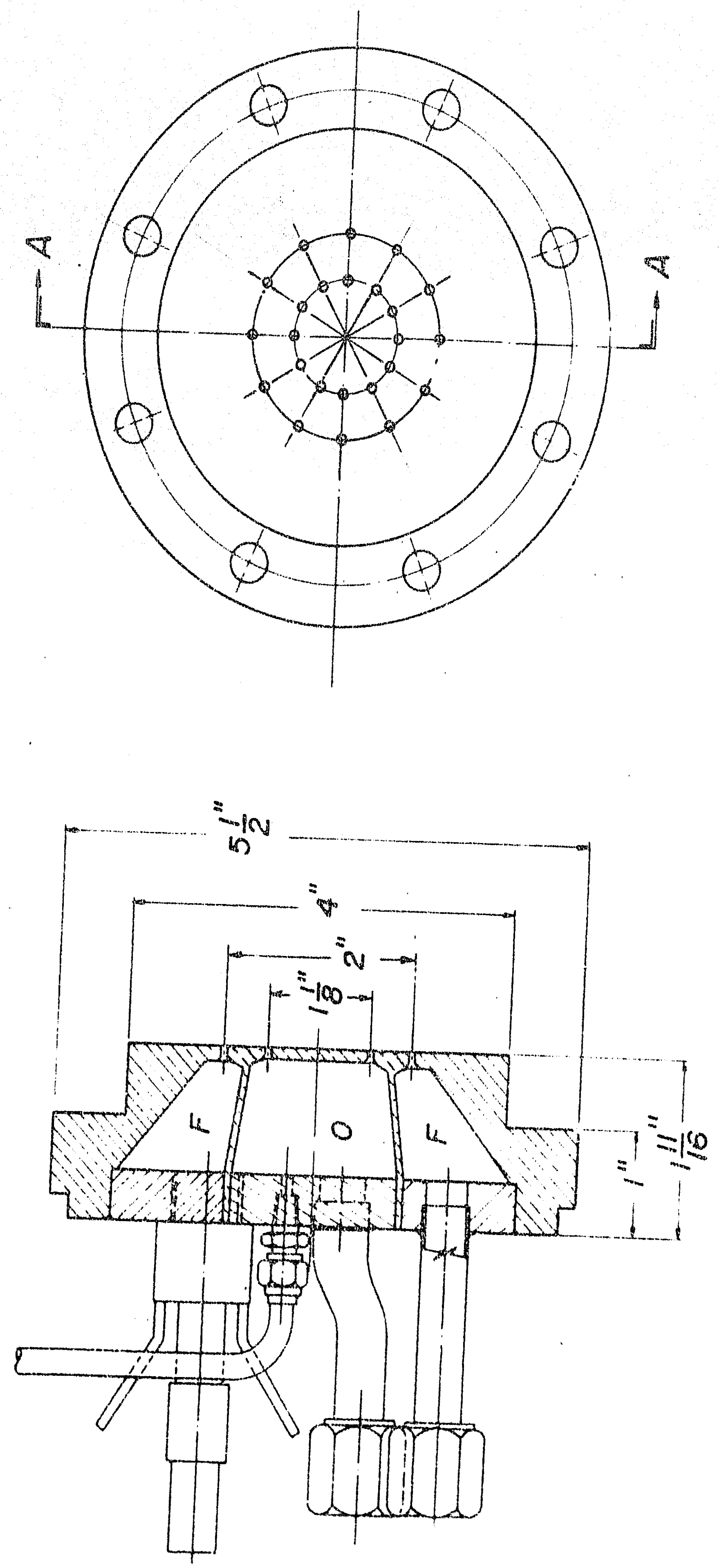
Figure 4



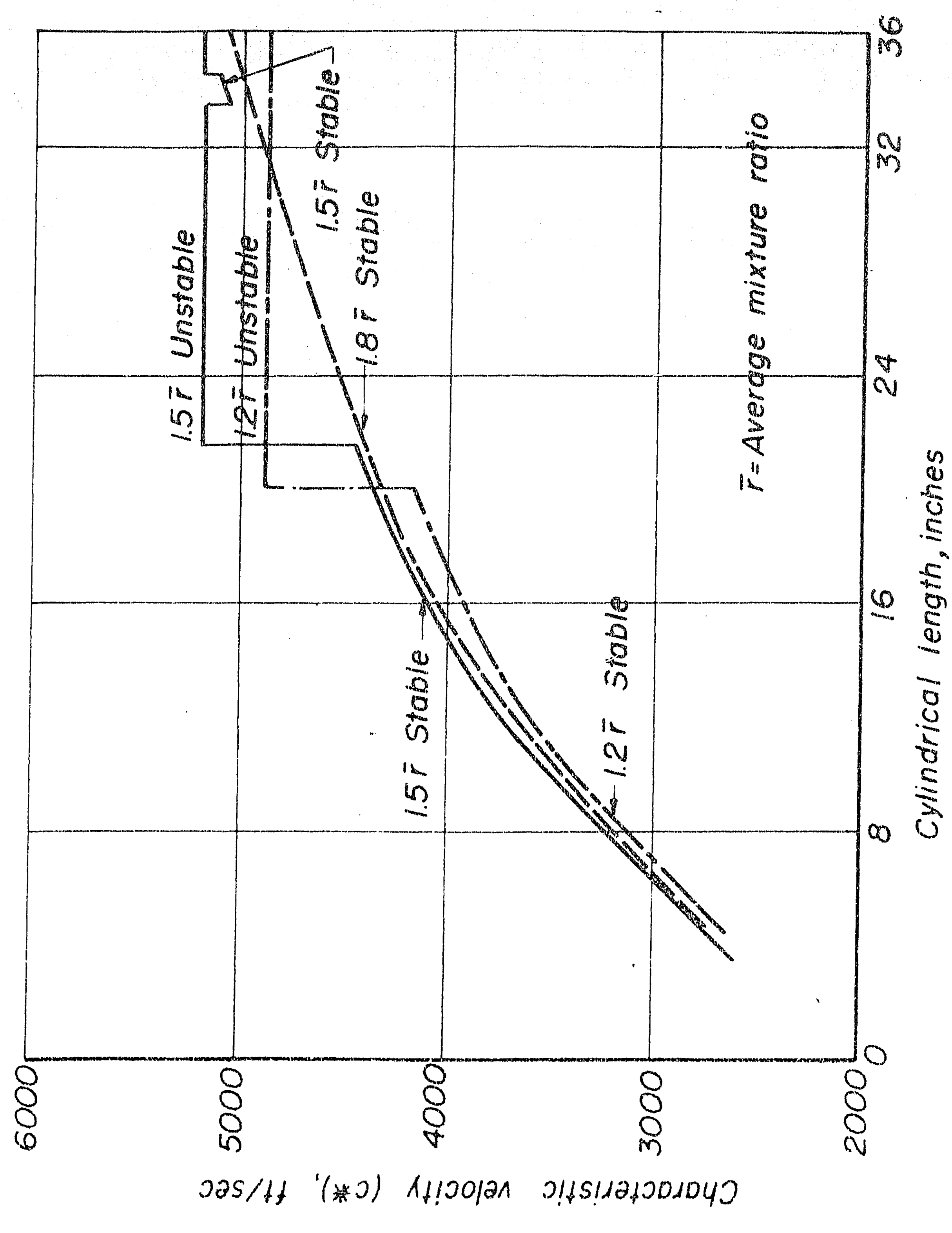
Recirculation region with
low mixture ratio operation
High mixture ratio
resultant streams
3" Station
1" Station
Low mixture ratio
resultant streams

Figure 5




Section AA

Showerhead injector



Characteristic velocity (c*), ft/sec
6000
5000
4000
3000
2000
0
8
16
24
32
36
Cylindrical length, inches
r̄ = Average mixture ratio
1.5 r̄ Unstable
1.2 r̄ Unstable
1.5 r̄ Stable
1.8 r̄ Stable
1.5 r̄ Stable
1.2 r̄ Stable

Figure 7

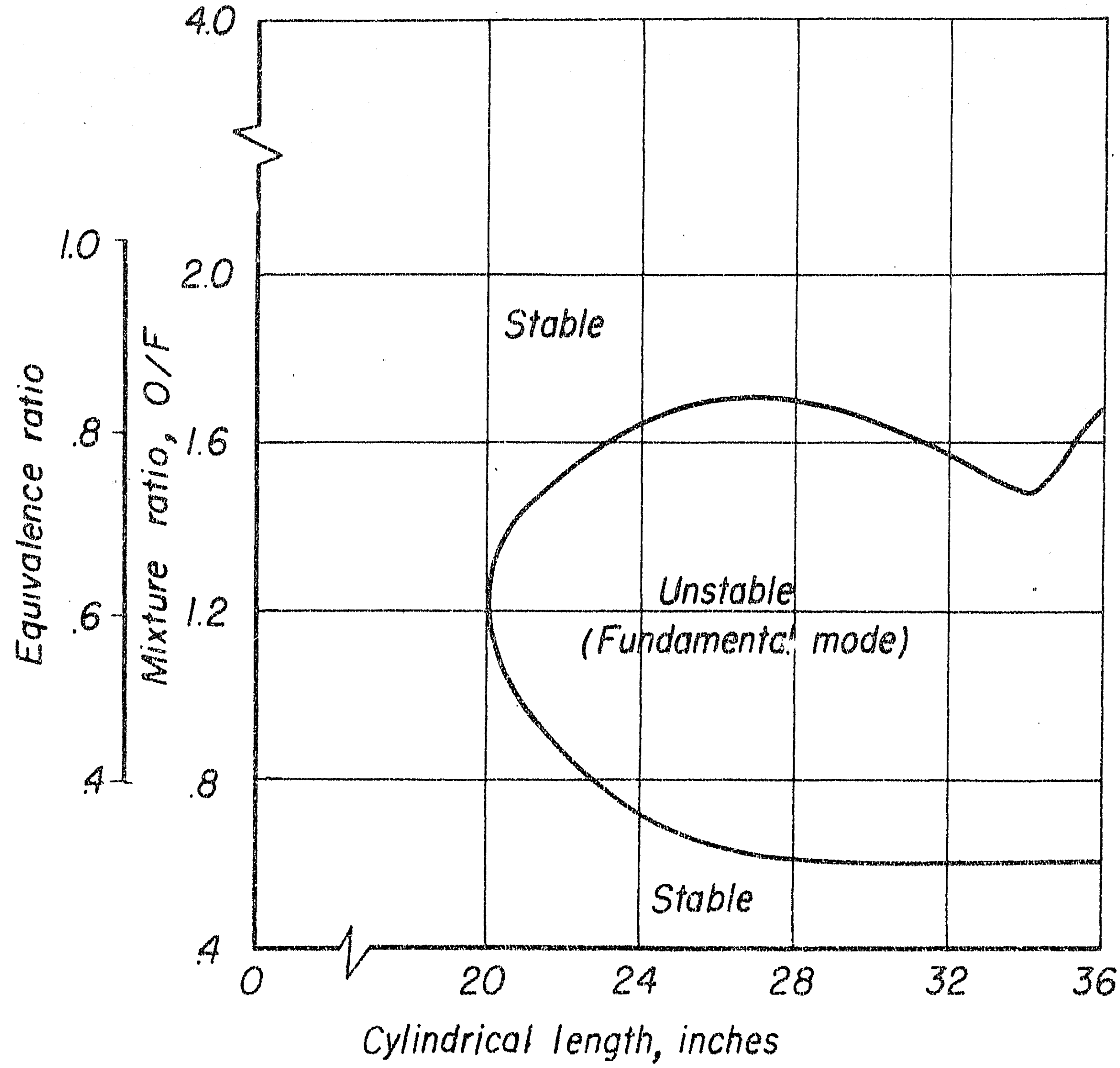
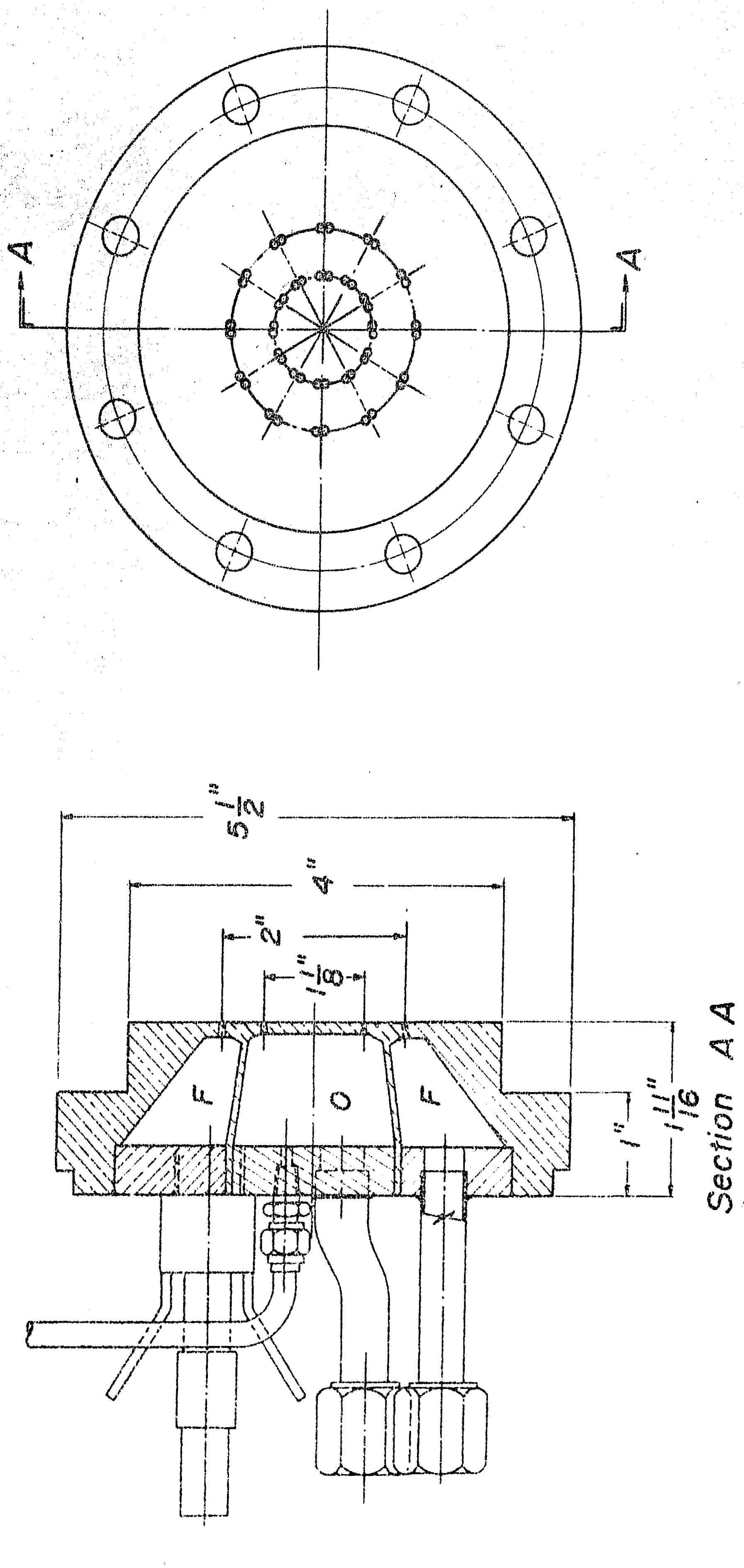

4.0
1.0
2.0
Stable
Equivalence ratio
Mixture ratio, O/F
.8
1.6
.6
1.2
Unstable
(Fundamental mode)
.4
.8
Stable
.4
0
20
24
28
32
36
Cylindrical length, inches

Figure 8




Like-on-like injector

Figure 9

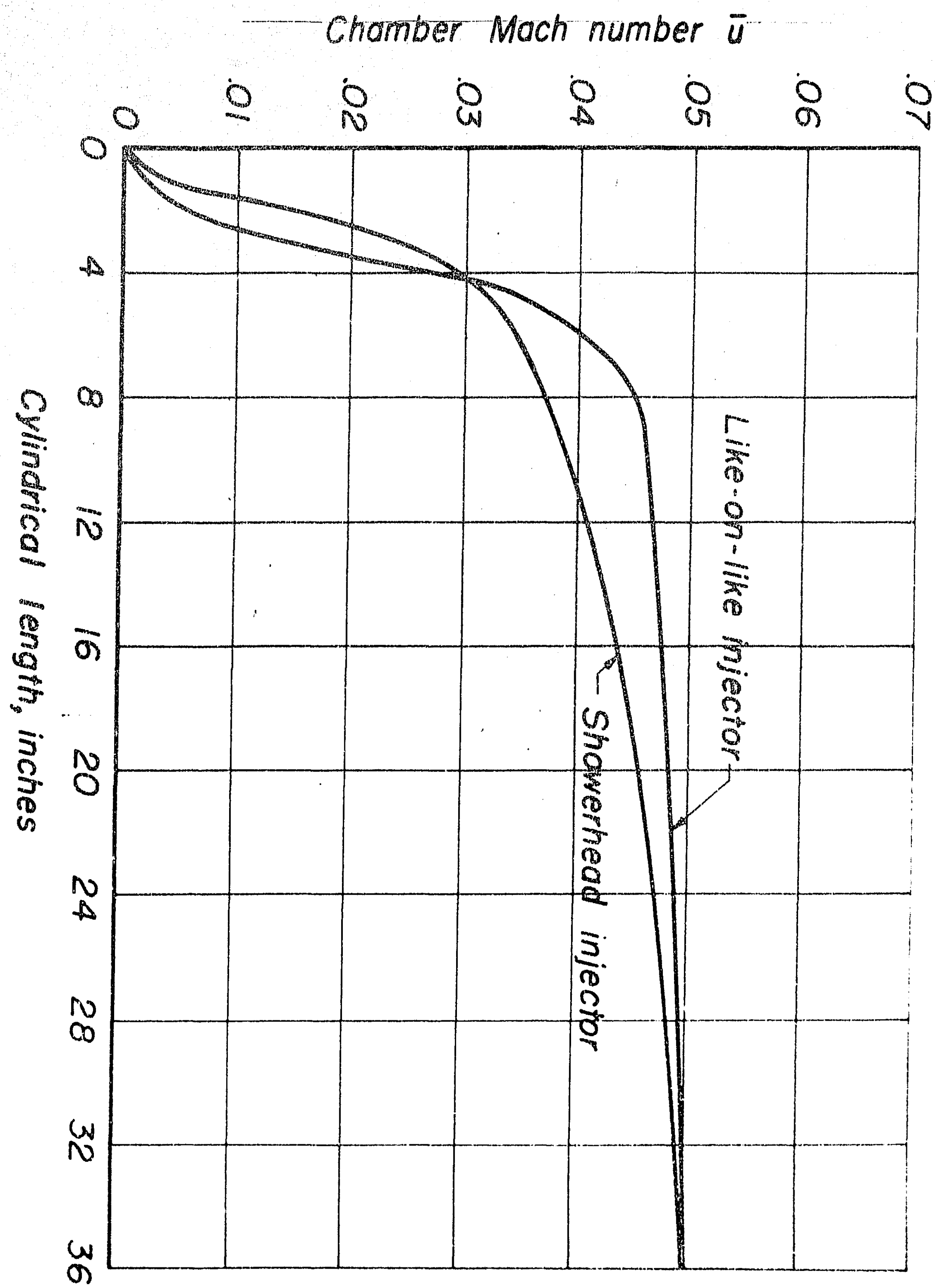
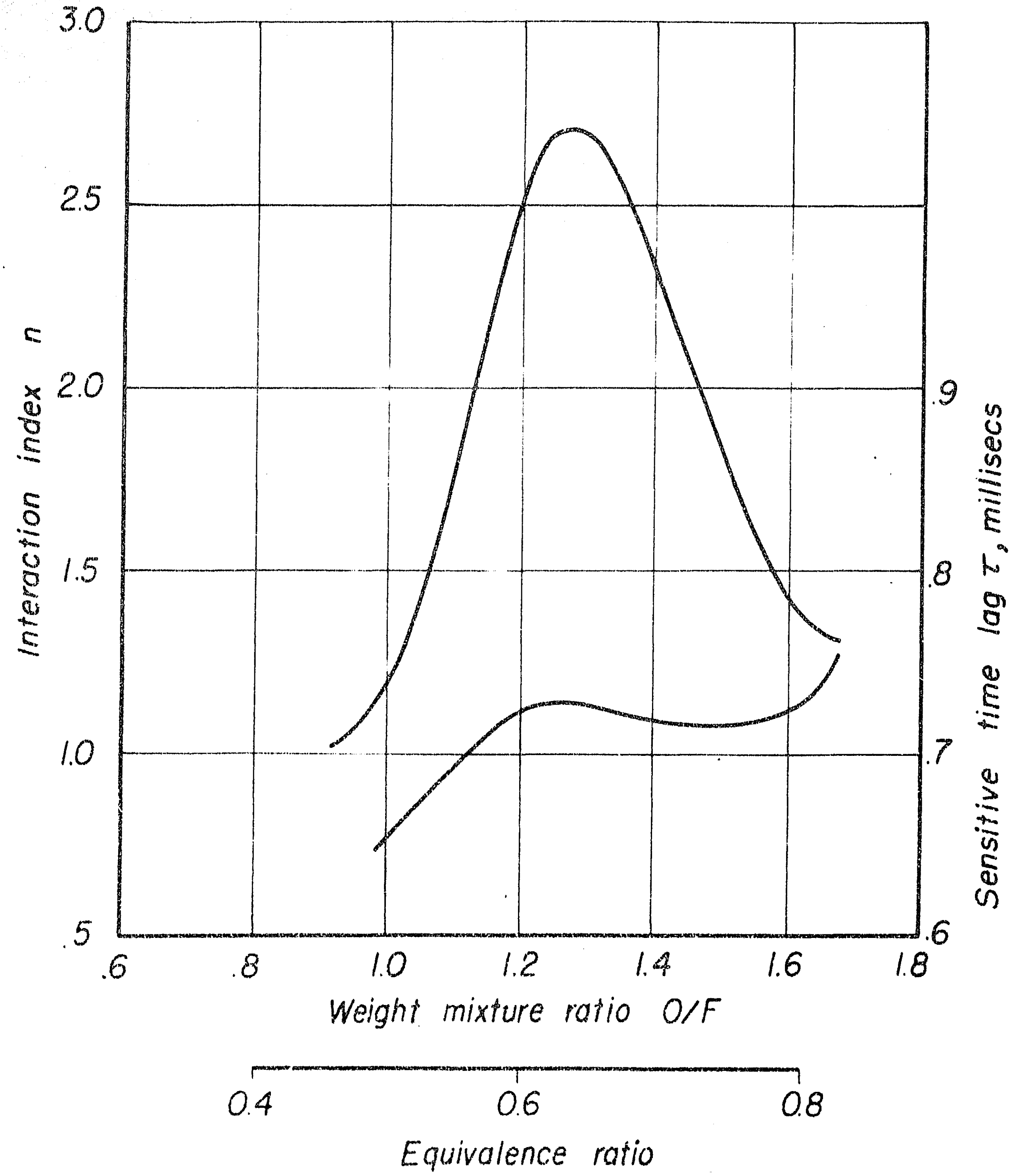
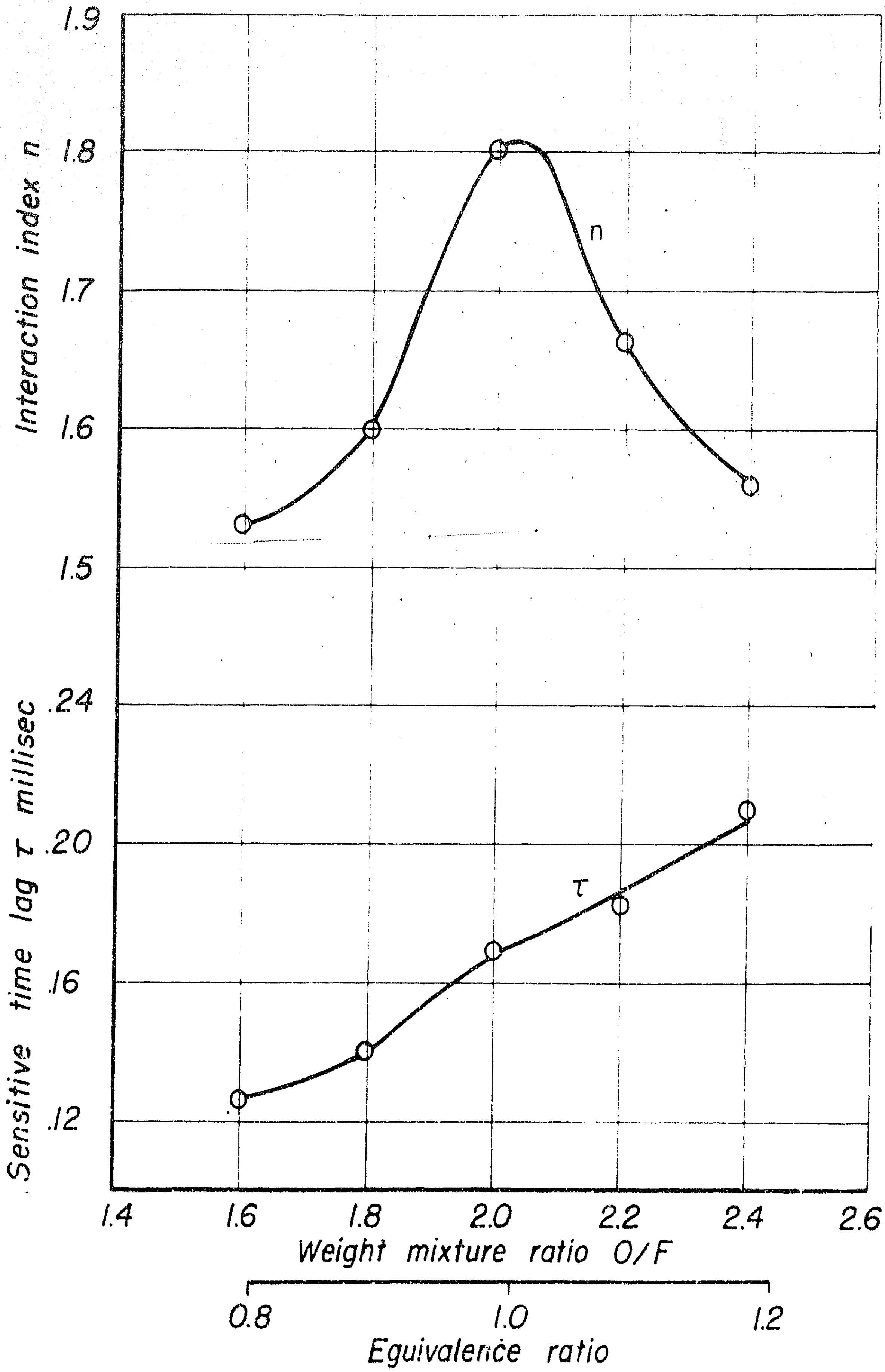

Chamber Mach number ū
0
.01
.02
.03
.04
.05
.06
.07
0
4
8
12
16
20
24
28
32
36
Cylindrical length, inches
Like-on-like injector
Showerhead injector

Figure 10



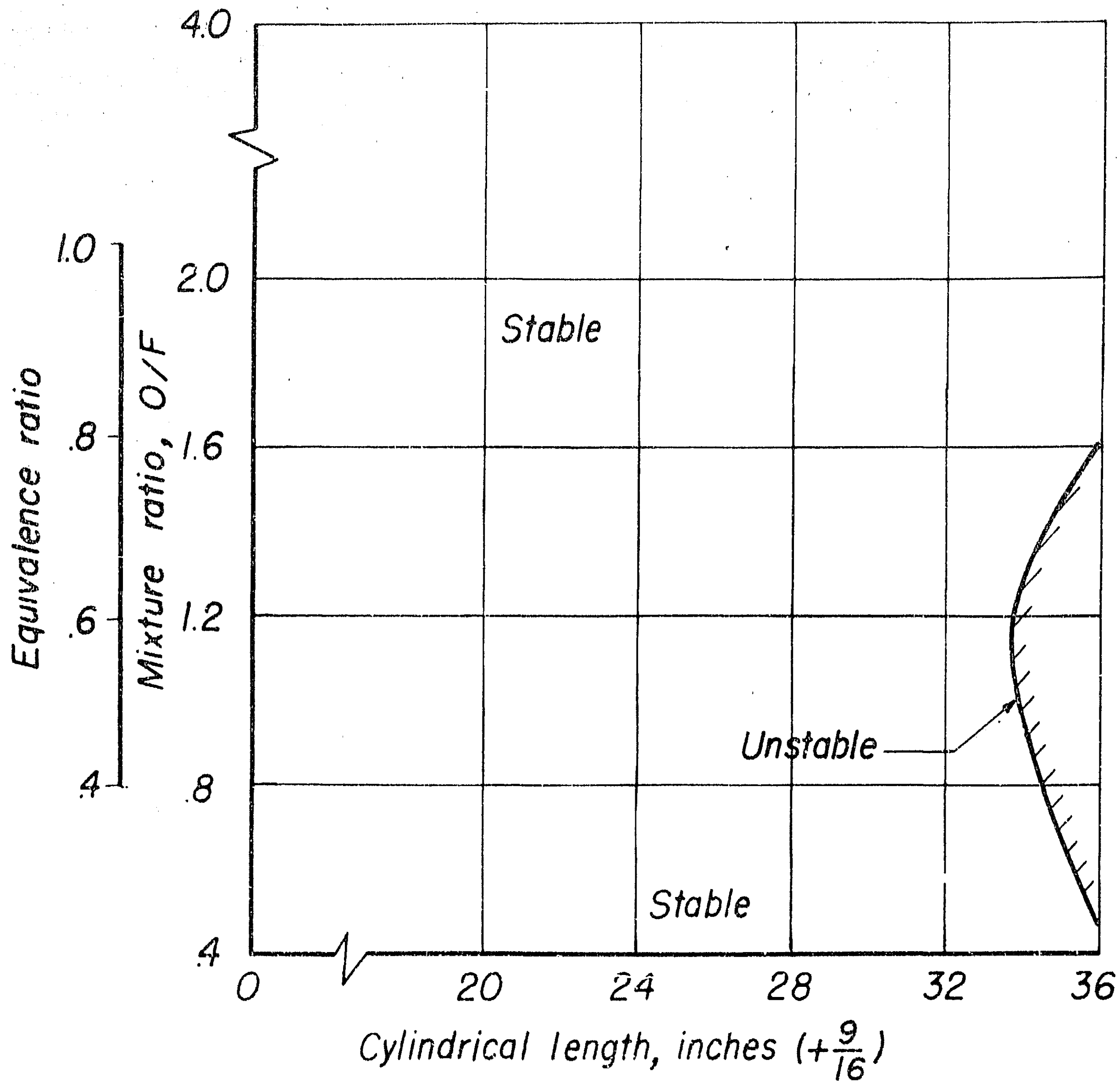
3.0
2.5
2.0
1.5
1.0
.5
Interaction index n
.9
.8
.7
.6
Sensitive time lag τ, millisecs
.6
.8
1.0
1.2
1.4
1.6
1.8
Weight mixture ratio O/F
0.4
0.6
0.8
Equivalence ratio

Figure 11



1.9
1.8
1.7
1.6
1.5
Interaction index n
n
.24
.20
.16
.12
Sensitive time lag τ millisec
τ
1.4
1.6
1.8
2.0
2.2
2.4
2.6
Weight mixture ratio O/F
0.8
1.0
1.2
Eguivalence ratio

Figure 12


4.0
2.0
1.6
1.2
.8
.4
1.0
.8
.6
.4
Equivalence ratio
Mixture ratio, O/F
Stable
Unstable
Stable
0
20
24
28
32
36
Cylindrical length, inches (+9/16)